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A New Mission Concept for the Space Infrared Telescope Facility

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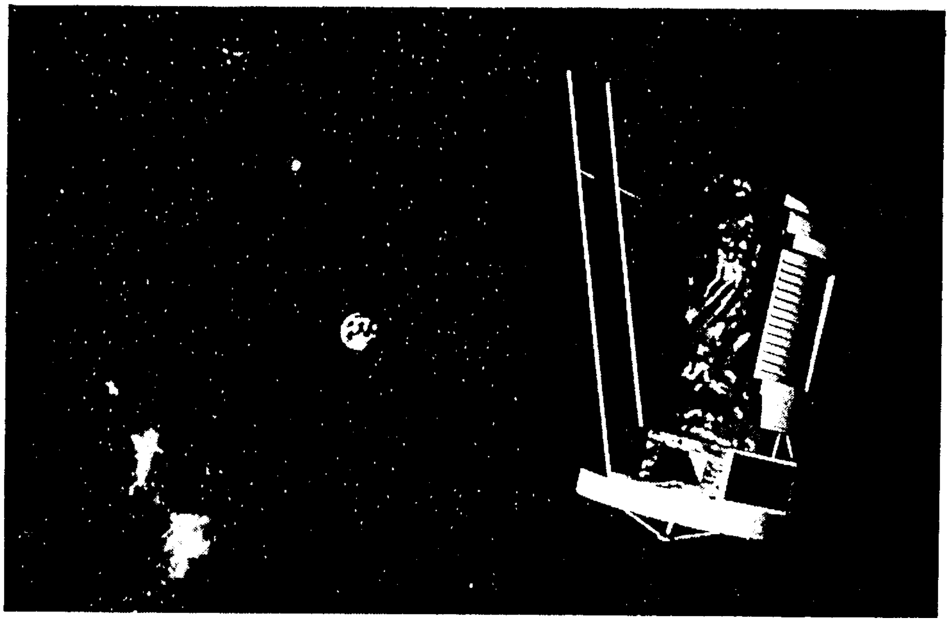

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A New Mission Concept for the Space infrared Telescope Facility

Johnny H. Kwok'

This paper describes a new mission concept for the Space Infrared Telescope Facility (SIRTF). In this concept, the observatory is launched with just enough energy to escape Earth's gravity. The Earth escape trajectory is equivalent to a 1-AU solar orbit where the observatory basically will fly in formation with the Earth. Without any propulsion capability, the observatory will drift slowly away from the Earth at a rate of 0.1 AU per year. This paper compares this mission concept with the high Earth orbit and the libration point mission concepts and describes how this concept has allowed the use of the much smaller and cheaper Atlas IIAS launch vehicle instead of the Titan IV launch vehicle. The paper gives some details in the selection of the launch trajectory, the optimization of the drift rate, the sensitivity of the drift rate to injection error, and the impact of this mission concept on the design of the telecommunications system, the solar panel, the aperture shade, and the observational profile.

INTRODUCTION

The Space Infrared Telescope Facility (SIRTF) will be a one-meter-class cryogenically cooled infrared astronomy observatory planned to be launched around the year 2001, SIRTF will be the infrared component of NASA's family of Great Observatories, which includes the Hubble Space Telescope (11ST), the Gamma Ray observatory (GRO), and the Advanced X-ray Astrophysics Facility (AXAF). SIRTF will cover the entire spectral region from 2 to 200 μm and thus uniquely encompasses the important "cosmic window, " the deep minimum in the natural background radiation located around 3.5 μm. This window provides deep views of the early Universe. SIRTF's optical system will provide diffraction-limited images at wavelengths longward of 3 μm, and the pointing accuracy and stability are matched to the < 1 arcsec image diameter at 3 μm. SIRTF will use the latest technological advances in large format infrared detectors to

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further the scientific achievements established by two predecessor cryogenic space systems, NASA's Infrared Astronomical Satellite (IRAS) and Cosmic Background Explorer (COBE) missions. The scientific importance and technical and programmatic readiness of SIRTF have been recognized by the 1991 report of the National Research Council's Astronomy and Astrophysics Survey Committee that recently identified SIRTF as the highest priority major new initiative in all of astronomy for the coming decade[1].

The original concept of SIRTF in the early 80's was to usc the Space Transportation System (Shuttle) to launch the observatory into a 900 km orbit. The mission lifetime was to be 10 years and required 1 to 2 Shuttle servicing missions. In late 1988, an alternative mission concept was conducted based on a 100,000 km altitude orbit launched by the new Titan IV/Centaur with the upgraded Solid Rocket Motor (SRMU). Mission lifetime was reduced to 5 years but, because of the 2 to 3-fold efficiency gain and the improved radiation environment at the high altitude, there was an overall improvement on science return. In the summer of 1989, the new concept was adopted by NASA and the science community to become the baseline for SIRTF[2]. In the fall of 1991, it became apparent to NASA and the SIRTF project that SIRTF as was conceived and designed was not commensurable with the fiscal and programmatic climate. In response to the guidance" given to the project by both NASA and Congress, the SIRTF scientific and engineering teams began to develop an alternate mission that retains much of the fundamental scientific importance and promise of the original SIRTF concept while permitting significant cost savings. The engineering team and the instrument teams were charged to redefine the instruments, the mission, the telescope, the spacecraft subsystems, and the operations concepts to minimize cost and complexity. The work began in March of 1992 and the SIRTF teams emerged in July with a completely new design.

The high Earth orbit was abandoned in favor of a solar orbit. The mission lifetime requirement was reduced from 5 years to 3 years. The instruments were simplified. The optical assembly of the telescope was reduced in size by using a faster f number. The 3-axis articulating secondary became fixed. A cold fine guidance sensor was eliminated and replaced by a much simpler quad sensor similar to the one used by the European Infrared Space Observatory (1S0). The thermal design of the telescope was optimized to take advantage of the solar orbit. With all these changes, it became feasible to use the Atlas IIAS launch vehicle instead of the Titan IV launch vehicle.

Figure 1 compares the size of the new observatory with the old Titan IV version, IRAS, and 1S0. The Titan SIRTF has an aperture of 92.4 cm and a 5 year lifetime. The new SIRTF observatory has an aperture of 85 cm and a design lifetime of 4.11 year (3 year is the minimum requirement). It is about the same size as IRAS but much smaller than 1S0. IRAS had a mission lifetime of 11 months and an aperture of 60 cm, and 1S0 has a planned mission lifetime of 18 months and an aperture of 70 cm.

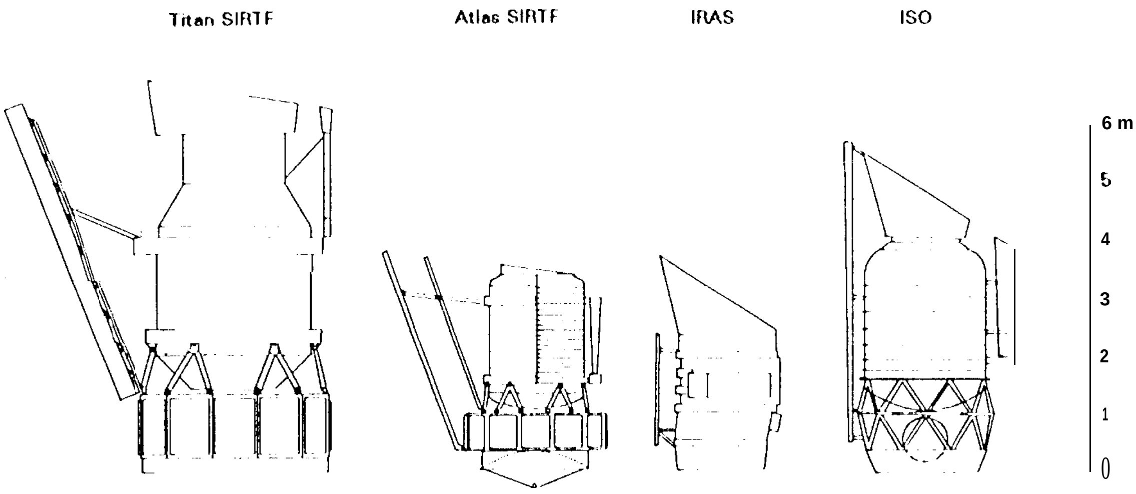

Fig. 1 Size comparison of infrared space telescopes

Orbit Options

One of the objectives in the process of downsizing and descoping SIRTF is to explore options of using some other orbit and launch vehicle.

The 1989 concept of SIRTF places the observatory in a 100,000 km altitude circular high Earth orbit (HEO) with an orbital period of 4 days. This orbit is a significant improvement over the shuttle launched low Earth orbit in terms of observational efficiency, radiation and thermal environment. The Titan IV/Centuar is the only U.S. launch vehicle that could deliver the old observatory to the 100,000 km orbit. The Centaur upper stage is designed to deliver payloads to a geosynchronous altitude of 36,000 km that has a transfer time of about 6 hours. The SIRTF HEO transfer requires about 19 hours. Consequently during the long coast, the Centaur requires additional batteries with associated electrical system modifications. There is also some concern over the ability of the Centaur engine to restart after the long coast because of the large ullage of the LOX tank. Some kind of mixer may have to be added to the LOX tank, All these modifications will add to the roughly $300M cost of the Titan vehicle. The current Titan IV/Centaur can deliver 5200 kg to the HEO, and 5700 kg with the Solid Rocket Motor Upgrade.

The next smaller US vehicle is the Atlas I] AS. The Atlas IIAS can deliver only about 1500 kg to the HEO. This would have been too much of a reduction, not to mention the problem with the long coast is still unresolved. The only way to use the Atlas is to find a new orbit that requires less launch vehicle energy to achieve. A high earth orbit at an altitude of 100,000 km requires a perigee transfer burn of about 2.93 km/s and an apogee circularization burn of about 1.28 km/s. These two burns are equivalent to an injection energy, C_3, of22 km^2/sec^2. By comparison, the C_3 required to send payloads to Venus Or Mars is about 10 km^2/sec^2. On the other hand, an escape orbit ($C3 \approx O$) from Earth, requires a single burn of 3.25 km/s. At $C_3 = O$ km^2/sec^2, the Atlas IIAS has a payload capability of about 2500 kg, a gain of 1000 kg over the I IEO. It became clear that, in order to use a smaller and cheaper launch vehicle, SIRTF has to abandon the HEO and use an escape orbit from Earth. An Earth escape orbit when viewed from the solar system is equivalent to a solar orbit. That is, the observatory will fly in formation with the Earth. However, without any propulsion, the telescope will drift slowly away from the Earth. The solar orbit further improves the radiation' and thermal environment, as well as viewing geometry since there is no more Earth and Moon avoidance constraints.

At one point, a libration point orbit was briefly considered for SIRTF. The L2 libration point is about 1.5 million kilometers from the dark side of the Earth along a line joining the Sun and the Earth. An object placed at this point experiences a balance between the Sun/Earth attraction and the centrifugal force. The L2 libration point is far enough from Earth that the energy required to reach that point is almost the same as an escape trajectory. This point would be a good place to place an infrared telescope since the Sun and the Earth will always be on the same side in the sky, and the distance from the Earth is small enough that omni antennas can still be used. However, the major drawback of the libration point orbit is that it requires a propulsion system and precise navigation to achieve, and that it requires continuous orbit maintenance throughout the mission. The propulsion system adds mass and complexity to the mission. Contamination of the sensitive optics of the telescope by propellants is another concern. On the other hand, adding a high gain antenna for the solar orbit option only adds about 10 kg to the total observatory mass. Consequently, the L2 option is discarded in favor of the solar orbit option. Table 1 summarizes the relative comparison among the three orbit options.

Table 1 Relative comparison of orbit options

	HEO	L2	SOLAR
maximum payload	1500 kg	2500 kg	2500 kg
upper stage modification	yes	no	no
propulsion and maneuver	no	yes	no
Earth-Moon avoidance	yes	no	no
Solar occultation	2 hr	no	no
navigation & tracking	simple	more complex	simplest
viewing efficiency	good	better	better
thermal load environment	good	better	better
aperture shade & baffle	large	small	small
telecommunications	low gain	medium gain	high gain

The Atlas IIAS Launch Vehicle

Currently, NASA is in the process of procuring a block of intermediate class launch vehicles. The current SIRTF mission assumes the eventual launch vehicle used will be similar in performance to the Atlas IIAS produced by General Dynamics. This section describes some characteristics of the Atlas I] AS.

The family of Atlas 11 vehicles are stretched versions of the Atlas I vehicles with a lengthened Atlas I Centaur upper stage. The Atlas 11A has an upgraded RI.- 10 engine on the Centaur upper stage. The Atlas IIAS has four Thiokol Castor IVA strap-on solid rocket motors, two ground-lit at launch and two air-lit after burnout of the first pair.

The Atlas HAS has not flown yet, but it has been chosen to fly the NASA Solar and Heliospheric Observatory (S01 IO) mission in 1995. In addition, there arc three other (non-NASA) launches prior to that mission.

Table 2 provides the breakdown of the current estimate of the Atlas IIAS performance. The performance simulation includes the Intelsat funded Block 1 upgrades and the standard large 14 ft diameter payload fairing. The launch site is the Eastern Space and Missile Center (ESMC). To maximize performance, a planar, direct ascent is used to achieve the Earth escape trajectory. The payload fairing is jettisoned at the standard free molecular heating rate of 854 W/m^2. The maximum dynamic pressure is 6160 N/m^2 (750 lb/ft^2). Based on these flight rules, the Atlas IIAS provides a payload capability of 2809 kg. This value reflects 109 kg withheld as Flight Performance Reserve (FPR). Additional penalties and reserves are provided in the table.

The Flight Performance Reserve (FPR) is the propellant reserved in the upper stage to ensure a soft-command shutdown to attain desired injection conditions in the event of lower than normal (up to 3σ) launch vehicle subsystem performance. Sometimes, the FPR is also referred to as Mission Required Margin (MRM) or Target Required Margin (TRM). in the case of SIRTF, it may be possible to reduce the FPR in the future by relaxing the requirement for an accurate orbital injection state. This is possible because, unlike interplanetary missions, the solar orbit can drift freely in space.

The Launch Vehicle Contingency (LVC) reserve is held as upper stage propellant mass for potential design changes/uncertainties that adversely affect performance. These changes/uncertainties are typically hardware related for immature unproved (no flights) launch vehicles. The LVC is held by the launch vehicle contractor, in the case for the Atlas I] AS, General Dynamics Space Systems.

The Launch Vehicle Mission Peculiar (LVMP) reserve is the performance hit incurred due to mission peculiar hardware changes to the launch vehicle needed to satisfy mission integration requirements, An estimate of the LVMP for SIRTF is based on the current SOHO mission.

The launch vehicle manager's reserve is NASA's launch vehicle reserve intended to cover unanticipated performance changes between contract signing and launch. This reserve is generally released at different phases of the mission and becomes zero at launch.

A 30 minute launch window is currently assumed for SIRTF. There is a performance penalty associated with targeting to an inertial target from a rotating earth over a 30 minute launch window. The Atlas IIAS has continuously variable launch azimuth capability between 94° and 112°. To accommodate the 30 minute launch window, the launch azimuth varies from 94° to 99°. The choice of a 30 minute window is for

Table 2 Launch Vehicle Performance Summary

	payload mass (kg)
Atlas IIAS payload capability (C_3 = O)	2809
launch vehicle contingency (1 ,VC)	-125
mission peculiar performance penalty (est.)	-45
1 ,V manager reserve	-125
Launch window penalty (30 rein)	-30
injection energy reserve (C_3 = 0.5)	-25
Performance Recommendation	2459

preliminary planning only. It is possible to reduce the window when more detailed analyses can be carried out in the future.

The above performance is evaluated for a theoretical escape orbit with injection energy C_3 = O km^2/sec^2. With lunar and solar perturbations, a more optimal C_3 value of 0.5 km^2/sec^2 is desired as will be shown later

In summary, an observatory mass of 2459 kg with an appropriate amount of contingency is recommended.

INJECTION DESIGN

in the context oftwo-body orbital dynamics (the two bodies being the Earth and the spacecraft), an escape trajectory from Earth means that the spacecraft velocity relative to Earth approaches zero when the distance from Earth approaches infinity. That means when the spacecraft is far enough away from Earth, it will have the same heliocentric velocity as the Earth in the solar system. If this was true, the resulting heliocentric orbit would look the same no matter which direction we inject the observatory. However, the escape orbit is severely perturbed by the Sun as the spacecraft moves away from the Earth. Because of these perturbations and depending on the direction of injection, an injection energy of zero may not result in an escape trajectory.

An initial study was performed by approximating the Sun-Earth-observatory with the planar restricted three-body dynamical system to search for proper escape trajectories By assuming a tangential burn in a planar circular parking orbit, the injection energy and the injection point arc varied parametrically. ‘I’he maximum drift distance is recorded. It was found that there are two classes of escape trajectories that give minimal distance from the Earth. One class has the observatory leading the Earth with the injection point on the far side (near midnight) of the Earth relative to the Sun (figure 2). The other class has the observatory trailing the Earth with the injection point on the sun lit side (near noon). These findings are verified by high

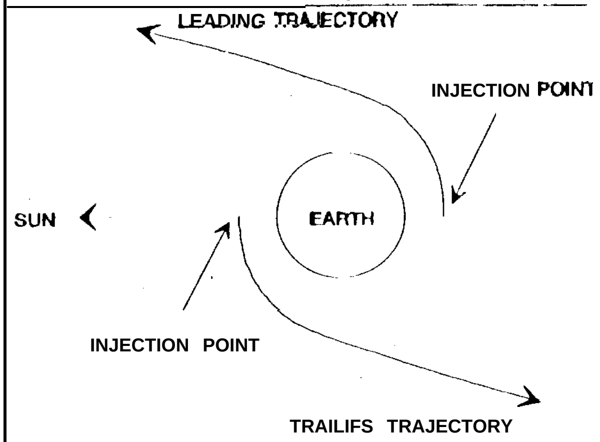

Fig. 2 Injection point geometry

precision trajectory models.

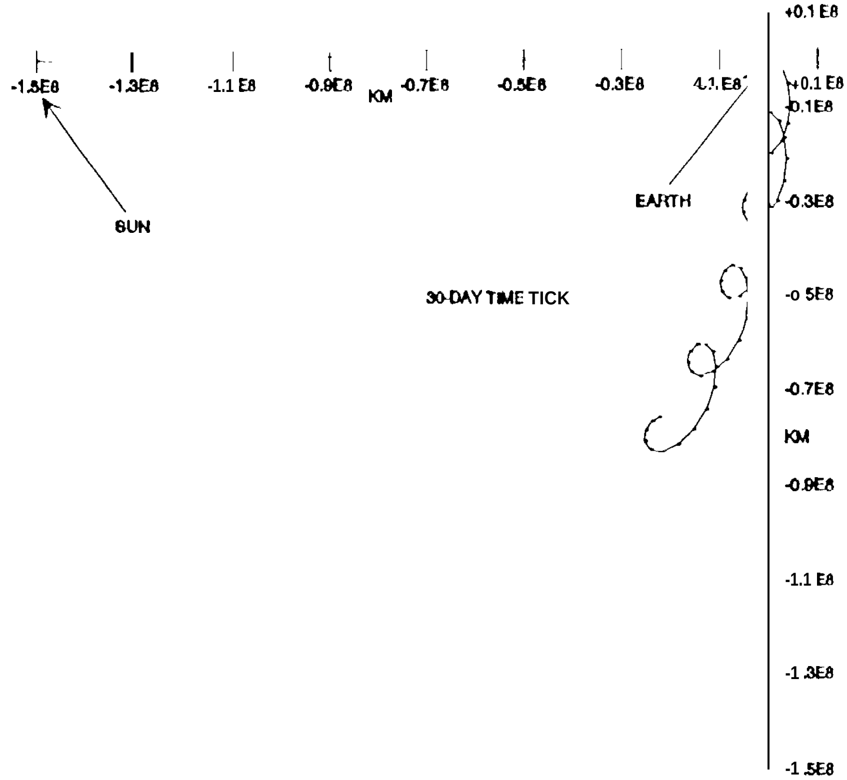

Fig. 3 5-year trailing trajectory

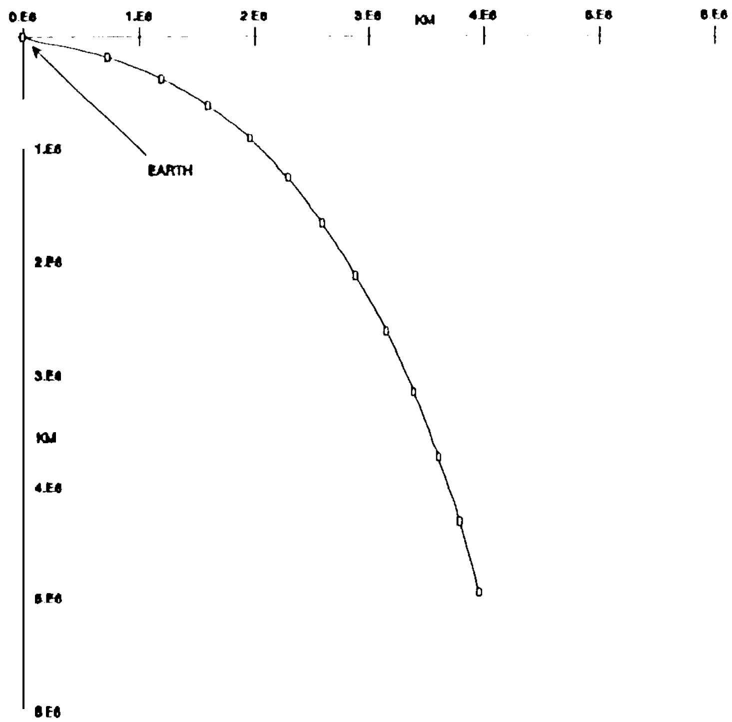

Fig. 4 Near Earth trailing trajectory

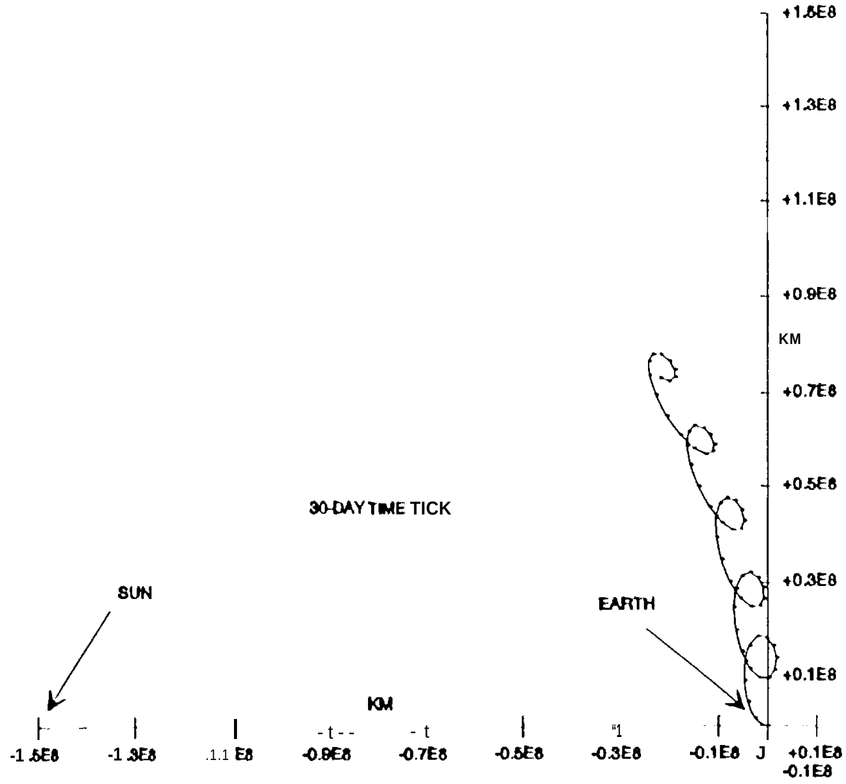

Fig. S S-year leading trajectory

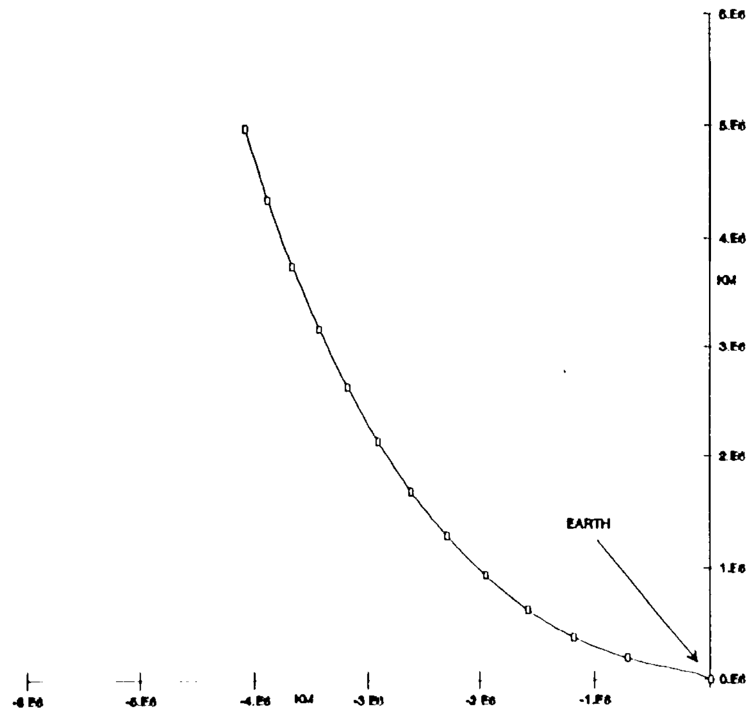

Fig. 6 Near Earth leading trajectory

Figure 3 depicts a representative trailing solar orbit for S years. The orbit is plotted relative to a fixed Sun-Earth line (the x-axis). The maximum distance from the Earth at the end of 5 years is about 0.55 AU (82 million km). This solar orbit is more eccentric than the Earth's orbit. Therefore, the observatory appears to move towards the Earth at perihelion and away from Earth at aphelion. Figure 4 shows the geometry of the orbit for the first 60 days after launch. Figure 5 shows a sample 5-year leading trajectory and figure 6 shows that trajectory for the first 60 days. The leading trajectory is rejected for three reasons. First, the injection point is near midnight which means that the ascent, injection, and portion of the near Earth trajectory will be in shadow. This will place additional power storage requirements on the observatory. Second, the near Earth trajectory places the observatory between the Sun and the Earth which complicates the initial check-out and calibration of the telescope and instruments. The last reason is the most compelling and is related to the telecom system design. The leading trajectory will not allow the use of the high gain antenna (HGA) during the near Earth trajectory phase. To point the HGA towards the Earth would violate the Sun avoidance constraint of the telescope. More discussion will be provided later.

MINIMIZING THE DRIFT RATE

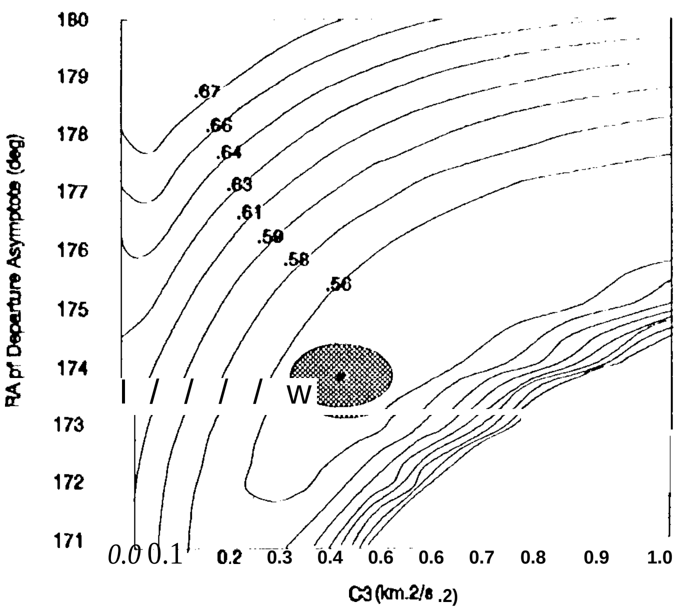

Fig. 7 Contours of maximum distance after 5 years

As mentioned earlier, the injection conditions of the escape trajectory is crucial in determining the shape of the solar orbit. Fig. 7 plots contours of the maximum distance of the trailing trajectories after S years as a function of injection energy and the right ascension of the outgoing asymptote (that is the direction of the velocity vector at infinity relative to Earth, V_∞). The reference trajectory selected has a C_3 of about 0.4 km^2/sec^2 and a right ascension of about 173.5°. The shaded area indicates an expected 3σ dispersion from the Atlas Centaur upper stage. The reference trajectory is selected to minimize the maximum distance after 5 years, the injection energy (C_3), and the sensitivity due to dispersion of the injection burn.

High precision simulation of the optimal reference trajectory indicates that its characteristics will vary slightly as a function of the launch time of the year. Therefore, a complete optimization should be performed when a definite launch period is selected. Also, the dispersion of the upper stage shown above is only a rough estimate. A complete covariance analysis has to be performed in the future to ensure the proper selection of the reference trajectory, Also, the launch vehicle performance assumes a direct ascent profile. A direct ascent is only possible for a narrow range of outgoing asymptote. Otherwise, a short coast is required to allow the upper stage to coast to the proper injection point in the parking orbit. Therefore, the final optimization process must take into account the tradeoff between the penalty due to a parking orbit coast versus a non-optimal trajectory (in terms of maximum distance after 5 years).

STATIONKEEPING OPTION

The current SIRTF spacecraft concept uses helium cold gas for the attitude control subsystem to perform momentum management of the reaction wheels. A study was performed to augment the current propulsion system to allow some amount of velocity management in order to keep SIRTF from continually drilling away from Earth. The study showed that for a SO m/s AV capability, the mass increase is about 500 kg when a helium cold gas system is used. This is obviously a poor trade compared to the mass of a high gain antenna system. If contamination is not an issue, a hydrazine system still weighs about 80 kg. This is still a poor trade compared to a high gain antenna system which adds less than 10 kg to the observatory.

It was suggested that if switches arc available at the outlet valves of the vent lines, one can use the helium boiloff to impart some amount of AV to the trailing trajectory. As it turns out, the available helium venting is equivalent to about 25 m/s of AV. A study was performed whereby this amount of impulse is imparted in a constant direction along the trailing trajectory over a period of 5 years. The end result is that the maximum distance from Earth after 5 years is reduced by only 5%. Given the expenses of adding switching valves along the venting lines and the control and operational complexity, the stationkeeping option is not recommended for SIRTF.

OBSERVATIONAL ATTITUDES

The solar orbit affords a dramatic reduction in the aperture shade of the observatory. For Earth orbiting observatories (IRAS, 1S0, HST), an aperture shade is required to avoid radiation from the Sun and the Earth from entering the telescope. The original Shuttle version of SIRTF uses a 60/60 aperture shade. A 60/60 aperture shade means that the design of the shade allows the line of sight of the telescope to be pointed up to 60° from the direction of the Sun and the Earth. The SIRTF design for the 100,000 km orbit uses an 80/80 aperture shade (see figure 1). With the solar orbit, the aperture shade can actually be eliminated. However, without an aperture shade, the telescope can only point up to 90° from the Sun, thus missing the regions at the ecliptic poles. Consequently, a decision is made to have a sun avoidance angle of 8S°, allowing a 5° cone at the ecliptic poles that can be viewed at any time during the mission for calibration and safehold.

Past spaceborne infrared telescope designs all used a fixed solar panel that is parallel to the telescope (IRAS, 1S0). In order to maximize the sky coverage and minimize the size of the solar panel, SIRTF uses a tilted solar panel. Figure 8 illustrates the sky coverage with a solar panel tilted 20° from the telescope. The solar panel is sized to provide nominal power with incidence angle of ± 65°. Now the telescope can be pointed -I S° and .4S° from the ecliptic pole with nominal power from the solar panel, giving nearly 40% of sky coverage. only 14 .7 % of the sky is considered off-sun pointing where power may have to be supplemented by the batteries

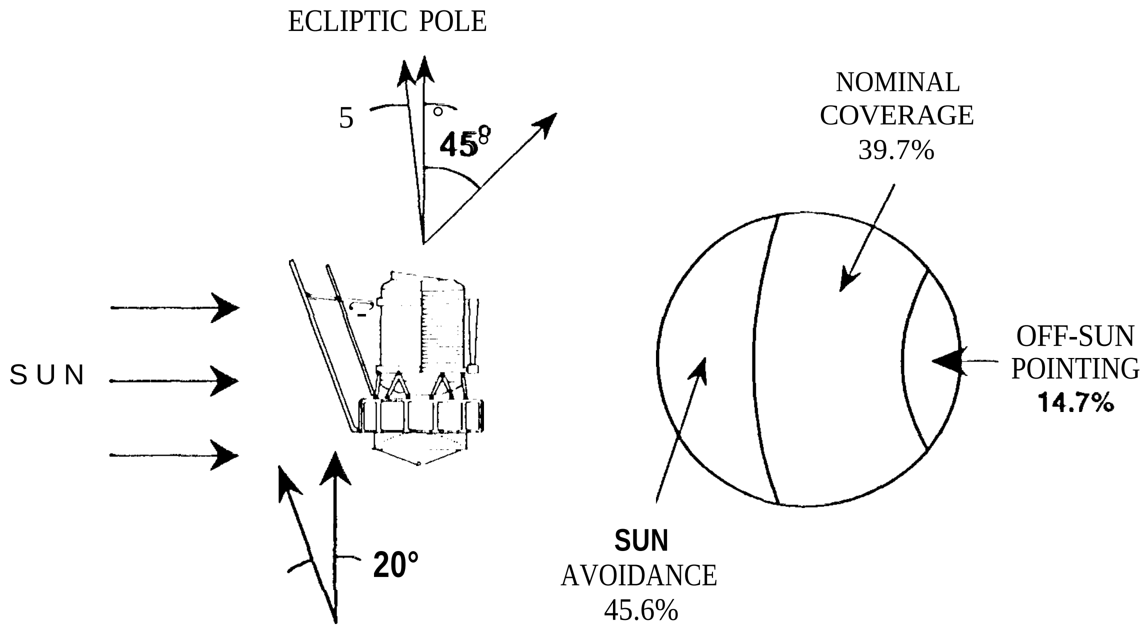

fig. 8 Observational attitudes

One additional advantage of the tilted solar panel is that it provides a larger gap between the back of the panel and the telescope allowing more of the heat from the solar panel to radiate into space. This situation is improved even more with an addition of a tilted heat shield between the solar panel and the telescope. The heat shield is sized to allow ± 5° roll around the optical axis.

COMMUNICATIONS

"l'he disadvantage of the solar orbit as compared to an Earth orbit is that it requires a more capable communication subsystem. The most desirable communication subsystem from the point of view of science return would be one that does not interfere with observations. This would require a deployable and steerable antenna. However, when considering factors of risk, mass, complexity, and cost, a fixed antenna is more desirable. The design objective for a fixed antenna would be to minimize inefficiency.

The current planned DSN block V receivers permit a maximum downlink data processing rate of 2.2 Mbps (million bits per second). The deep space high efficiency HEF 34 meter and the 70 meter nets have both S- and X-band uplink and downlink where SIRTF has chosen X-band for higher link performance. A 1.5 meter Viking or similar antenna is chosen for the observatory. This is about the largest solid dish that can be packaged *in* the Atlas payload fairing and adapter without much complication. The transmit power is chosen to be 20 watts which allows the 2.2 Mbps downlink rate for 3 years with the 34M HEF net, and 5 years with the 70M net. Figure 9 shows the achievable data rate as a function of mission elapse time for the 34M HEF and the 70M nets for elevations of 10° and 25°. The current design assumes two data rates, 2.2 Mbps and 1.1 Mbps. The current planned averaged data taking rate over a 12 hour period is 45 kbps. With downlinking twice a day, the on-board storage requirement is 2 gigabits. At 2.2 Mbps, these data will be downlinked to the DSN in 15 minutes twice a day. Assuming an average slew to point the antenna to and away from the Earth is 90°, then the current Pointing and Control Subsystem (PCS) conceptual design will take about S minutes per slew. At 25 minutes per 12 hours, the inefficiency is 3.5% due to communication. The full antenna beam width at half power is about 1.6°. Throughout the mission lifetime, the telescope can make observations within this band along the ecliptic plane. If one plans for some amount of observations during the communication time, then the inefficiency can be reduced.

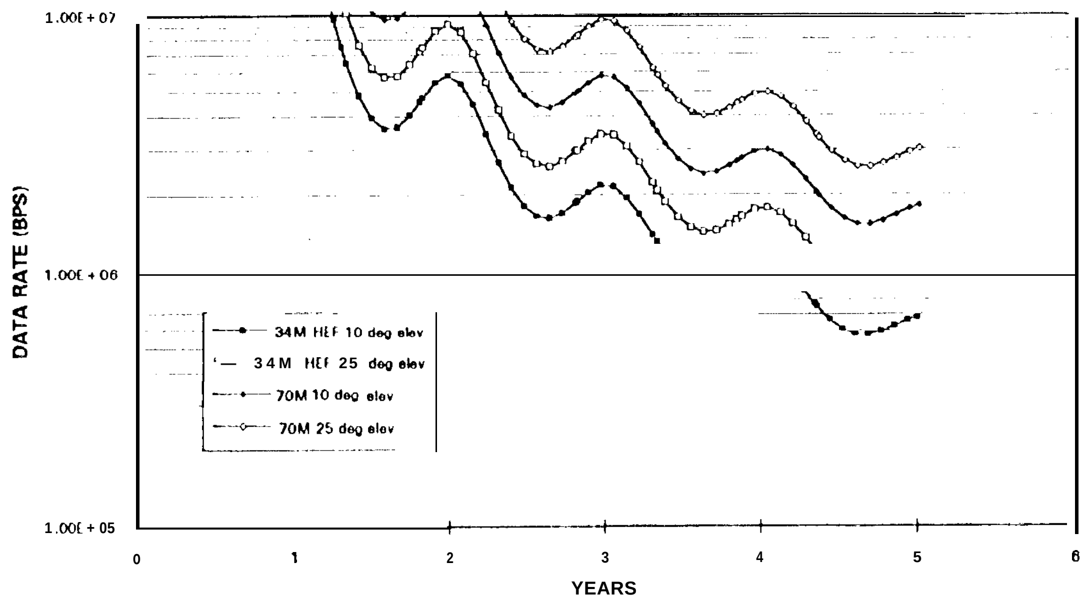

Fig. 9 Downlink data rate

The antenna is located at the bottom of the telescope as shown in figure 8. Locating the antenna to the side of the telescope will cause thermal problems and will require the telescope to roll more than 5° during the mission lifetime. In addition, the bottom mounted antenna allows its usage as early as 30 days after launch (see figure 4) with adequate solar incidence power and without violating thermal constraints. The leading orbit does not allow pointing the antenna at Earth early in the mission because the telescope will be pointed towards the Sun (see figure 6).

Up to 60 days after launch is the In-Orbit-Checkout (1OC) period. Because of the Earth location relative to the Sun during this period, it is not possible to point the high gain antenna at the Earth and maintain nominal power and satisfy thermal constraints, During this period, the observatory will communicate via one of the two X-band omni antennas, Because of the reduced link performance, data rate will be at 45 kbps. Continuous coverage from the DSN 34M HEF net is anticipated for the first 30 days. Between 30 to 60 days, 70M net will be used. It is also possible to communicate with Earth during this period using the HGA. The consequence is that the spacecraft may partially go on battery. This slightly off-sun pointing option for 30 minutes every 12 hours is well within the power budget of the battery and solar array design. Emergency telemetry will be possible through the OMNI at 40 bps throughout the mission.

Table 3 Uplink and downlink strategy

PERIOD	COMM LINK	DATA RATE	DSN COVERAGE
DOWNLINK			
L -L+ 30 days	OMNI	45 kbps	34HEF Continuous
L+30 - L+60 days	ohm] or OMNI and HGA off-sun	45 kbps 5 kbps 2.2. Mbps	70M Continuous 34HEF Continuous 34HEF two 30 min passes/day
L+60 days -2.25 yr	HGA	2.2 Mbps	34HEF, two 30 min passes/day
2.25 yr -5 yr	HGA	1.1 Mbps or 2.2 Mbps	34HEF, two 1 hour passes/day 70M, two 30 min passes/day
UPLINK			
L - 3 yr	OMNI	2 kbps	34HEF, onc 30 min pass/week
3 yr - 5 yr	OMNI	500 bps or 2 kbps	34 HEF, one 2-hour pass/week 70M, one 30 min pass/week

It is desirable that uplink command does not interfere with observations. Because of the much lower data rate for uplink, it is possible to use the OMN1 throughout the mission. With a desired commanding rate of once per week, a rough estimate of the required commands for 8 days of operations (a week plus one contingency day) is 200 kwords (16 bits). The current deep space transponder maximum command rate is 500 bps with a possibility of upgrade to 2 kbps. At 500 bps, a total of 2 hours is required pcr weekly load assuming 15°/0 overhead for the command communications protocol and headers. The 2 kbps rate is clearly more desirable. The current design assumes both rates are available.

Table 3 summarizes the uplink and downlink strategy throughout the mission.

TRACKING

Unlike Earth orbital and interplanetary missions where frequent tracking and precision orbit determination are required to achieve certain targets or orbit event execution, the only navigation requirement on the solar orbit option for SIRTF is to have enough knowledge of the observatory location for DSN antenna pointing. At X-band frequencies, acquisition by the 70 meter antenna requires angular accuracy's of approximately 0.015° for antenna pointing. A velocity accuracy of 70 m/s (2 Khz) is required to receive downlink signal. For the 34 meter antenna, the requirement is a factor 2 easier.

SIRTF will carry a standard NASA transponder with a capability of coherent two-way X-band Doppler. Two-way data may be acquired using either the high gain antenna or the omni. During the first 30 days of the mission, only the onmi antenna can be used because of telescope pointing constraint. After that, a tracking strategy for SIRTF is proposed based on taking advantage of the two daily telemetry passes in order to minimize the use of ground antenna resources. Essentially, "this strategy utilizes only two-way X-band Doppler. Each pass is extended for a duration of 2 hours. During the half hour telemetry segment, the high gain antenna is used to transmit two-way Doppler simultaneously with telemetry. For the remaining 90 minutes of the pass, the onmi antenna will be used for two-way coherent Doppler transmission. The two daily passes are assumed to be separated by approximately 12 hours and are supported by different DSN sites. Because of the rather relaxed navigation requirement, there is considerable flexibility in the above tracking strategy. As the results below indicate, once an initial state is established, it is possible to predict several months into the mission without further tracking data.

The results presented here are based on a covariance study using Doppler data to determine the best estimate of the spacecraft state. The effects of unmodeled dynamic acceleration errors due to solar pressure and due to the cryogenic venting system arc included in the computation of the statistics. The venting accelerations are treated as random stochastic acceleration (i.e., process noise) with a batch time of 6 hours. Random accelerations of 1.6 x 10-11 km/sec^2 are assumed for venting in all three directions; and an acceleration error of 10 percent is assumed for the solar pressure. This effectively assumes that venting accelerations cannot be modeled, Observational errors, such as station location errors, are neglected since their effect on the statistics should be comparatively small.

Three cases are investigated: a near-Earth case (distance 0.7 x 107 km) which represents the orbit in the early mission phase, a **maximum distance case (distance** 0.8 x 10s km) which is an end of the mission example, and a zero declination case. AH three cases assume 30-day tracking data and the errors (including the effects of unmodeled acceleration errors) are mapped 60 days beyond the end of the data arc. Table 4 summaries the (1σ) radial, angular, and range-rate errors for the three cases mapped 60 days beyond the end of the data arc.

Table 4 SIRTF predicted orbit determination errors

	Range error (km)	Angular error (deg)	Range rate error (cm/s)
Near earth	63	0.0013	2.3
Max distance	1170	0.0004	0.6
Zero declination	735	0.0024	3.8

OBSERVATORY DESIGN SUMMARY

The observatory consists of three major systems, the telescope, the spacecraft, and the payload. The detailed description of each of these systems is beyond the scope of this paper.

Table 5 Observatory mass breakdown

Spacecraft		778
Pointing Control	117	
Command and Data Handling	46	
Telecommunications	34	
Reaction Control	33	
Power	110	
Thermal Control	16	
Structure	356	
Mechanical Devices	9	
Cabling	50	
Attitude Gas	7	
Telescope		846
Cryostat Housing	388	
Optical Telescope Assembly	102	
Helium Tank	130	
Helium	133	
Ejectable Aperture Cover	69	
instrument Integration Hardware	22	
Quad Sensors	2	
Instrument		215
Contingency		627
Total	2466	

Table S provides a summary of the mass breakdown of the observatory. 'J"he payload consists of three instruments, the Infrared Array Camera (IRAC), the Multiband imaging Photometer for SIRTF (MIPS), and the Infrared Spectrograph (IRS). The designs of these instruments are evolving and the mass given in the table reflects an allocation to the payload. The mass contingency is composed of 35°/0 of the spacecraft and telescope dry mass, and 50% of the instrument mass. Table 6 gives a summary of the current spacecraft system design characteristics. The closed loop pointing accuracy is required by the spectrograph and is achieved by using the spectrograph peakup array. This is accomplished by first placing the target with an accuracy of 3 arcsec on the IRS peakup up array. The IRS is then responsible for identifying the target and determining the offset necessary to place the target in the center of the short wave spectrograph which has a slit size of 2 arcsec. Table 7 gives a summary of the current telescope system design characteristics. Table 8 gives the combined functional capability of the three instruments and the science they can achieve.

Table 6 Spacecraft system design characteristics

Average on-orbit power	720W
Pointing accuracy (open loop)	3"
Pointing accuracy (closed loop)	0.25”
Pointing stability	0.25" for 1000 sec
On-board memory (100% contingency)	4 gbits
Max downlink rate	2.2 Mbps
Attitude control	Reaction wheels
Momentum dumping	Helium gas
Telecom transmit power	20W
High gain antenna	1.5m fixed

‘J’able 7 Telescope system design characteristics

Telescope optics	Ritchey-Cretien
Aperture	85 cm
Aperture shade	85° solar avoidance
PRIMARY f/#	1.5
SYSTEM f/#	12
Wavelength	2.5-200 pm
Optical quality	diffraction limited at 3.5 pm
Unvignetted Field of View	26 arcmin
Central obscuration	10%
Helium	1000 liter
Bath temperature	1.22° K
Average Shell Temperature	77.1° K
Heat load to tank	18.1 mW
Helium mass flow rate	0.981 mg/sec
Designed lifetime	4.11 years

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Table 8 Summary of SIRTF instrument functionality

Wavelength range (µm)	Material, array size		Unique Science
Imaging		Field of view, pixel size	
2.5 -5.3	InSb, 256x256	5'×5', 1.2"	Protogalaxies
4-15	Si:As (IBC), 128x128	5'×5', 2.4"	Brown dwarfs
15-36	Si:Sb (IBC), 128x128	5'×5', 2.4"	Planetary debris disks
40-120	Ge:Ga, 32x32	5'×5', 9"	Interacting galaxies
120-200	Ge:Ga (stressed), 2x 16	0.6'×5', 19"	Early stages of star formation
Spectroscopic		Spectral Resolving Power	
4-12	Si:As (IBC), 128×128	1000 - 2000 (cross-dispersed)	Composition of interstellar material
12-40	Si:Sb (IBC), 128x128	1000-2000 (cross-dispersed)	Nature of Galactic nuclei
40-120	Ge:Ga, 4x32	1000-2000	Star formation
120-200	Ge:Ga (stressed), 2x16	500 -1000	Composition and energetics of interstellar clouds
Note: Low resolution spectroscopy (R ~ 150) from 2.5- 5 µm to be provided via grism in InSb camera with science focus on composition of solar system objects.			